Orbit Transfer Vehicle (OTV) Advanced Expander Cycle Engine Point Design Study

Contract NAS8-33574
Engine Data Summary
October 1980

Prepared For:
George C. Marshall Space Flight Center
National Aeronautics And Space Administration

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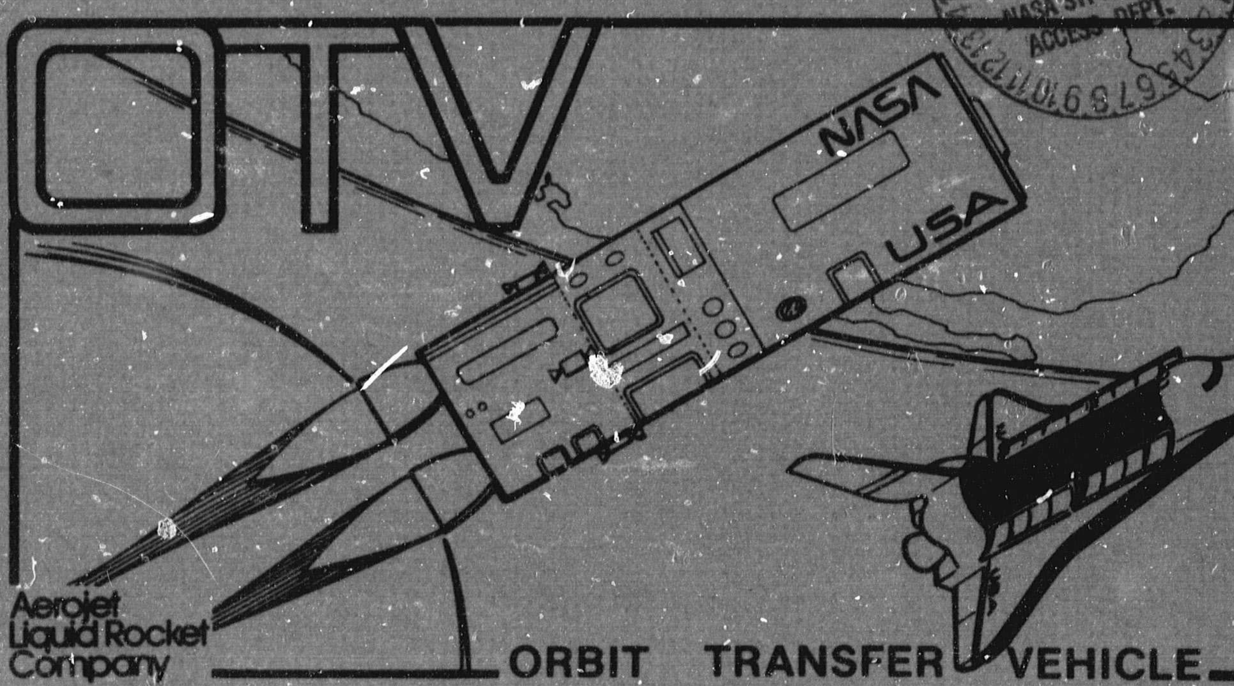

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ORBIT TRANSFER VEHICLE (OTV) ADVANCED EXPANDER CYCLE ENGINE POINT DESIGN STUDY

TASK VII

ENGINE DATA SUMMARY

Prepared For:

National Aeronautics and Space Administration
George C. Marshall Space Flight Center
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Contract NAS 8-33574

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FOREWORD

This report is submitted in partial fulfillment of Contract NAS 8-33574, Orbit Transfer Vehicle (OTV) Advanced Expander Cycle Engine Point Design Study. The data submitted herein was prepared in accordance with Task VII of the contract. This work was performed for the NASA/Marshall Space Flight Center by the Aerojet Liquid Rocket Company (ALRC).

The NASA/MSFC COR was Mr. D. H. Blount, the ALRC Program Manager was Mr. L. B. Bassham and the Study Manager was Mr. J. A. Mellish.

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I. INTRODUCTION

The Space Transportation System (STS) includes an Orbit Transfer Vehicle (OTV) that is carried into low Earth orbit by the Space Shuttle. The primary function of this OTV is to extend the STS operating regime beyond the Shuttle to include orbit plane changes, higher orbits, geosynchronous orbits and beyond. The NASA and the DoD have been studying various types of OTV's in recent years. Data have been accumulated from the analyses of the various concepts, operating modes and projected missions. With the inclusion of man in these transportation scenarios, it becomes necessary to reach for the <u>safest</u> and fully optimized propulsion stage.

The purpose of this study was to generate a performance optimized engine system design for a man-rated advanced LOX/Hydrogen expander cycle engine. This concept, originally conceived by ALRC on the OTV Phase A Contract (NAS 8-32999), exemplifies the merging of man-rated design issues coupled with high performance, reuseability and low development risk for the OTV engine.

II. SUMMARY

This Engine Data Summary was prepared to meet the Task VII requirements of the contract Statement of Work. The primary objective of this task was to prepare a document which summarizes the engine performance, weight, envelope and service life data and the engine and component layout drawing descriptions.

To accomplish the objective, most of the data contained herein is presented in tables, figures and drawings. The following categories of data for the Advanced Expander Cycle Engine are presented:

- Engine operating specification and pressure schedule
- Engine system layout drawing
- Major component layout drawings, including:
  - Thrust chamber and nozzle
  - Extendible nozzle actuating mechanism and seal
  - LOX turbopump
  - LOX boost pump
  - Hydrogen turbopump
  - Hydrogen boost pump
  - Propellant control valves

II, Summary (cont.)

- Engine performance and service life prediction

- Engine weight

- Engine envelope

The data presented in this report represents updates that are based upon current results from the design and analyses tasks performed under this contract. The engine and component designs have not been performed in sufficient detail to be considered as firm. Therefore, further iterations in the designs and data can be expected as the Advanced Expander Cycle Engine design matures.

III. ENGINE REQUIREMENTS AND OPERATING CHARACTERISTICS

The requirements for the Manned Orbit Transfer Vehicle (MOTV) engine were derived from numerous NASA in-house and contracted studies and are summarized on Table I. An engine thrust level of 15,000 lbF was selected for the OTV engine point design studies.

Based upon the results of design analyses, engine sensitivities, cycle optimization, and thrust chamber geometry optimization conducted in conjunction with both the OTV Phase A and this Point Design Study, an engine with the characteristics summarized in Table II was selected as a representative 1980 technology baseline. The data is presented for both nominal mixture ratio, 6.0, and off-design MR = 7.0 operation. The engine length with the extendible nozzle in the stowed position is 60 in. With the extendible nozzle deployed, the engine length is approximately 110 in.

The O_2/H_2 expander cycle engine uses a series turbine drive cycle which is shown on Figure 1. The engine uses hydraulically driven boost pumps, with the flow tapped off the main pump stages. Fuel flows from the pump discharge to the thrust chamber where 85% of the hydrogen flow is used to cool the slotted copper chamber in a single pass from an area ratio of 10.6:1 to the injector head-end. Fifteen (15) percent of the hydrogen is used to cool the tube bundle nozzle in two passes from an area ratio of 10.6:1 to the end of the fixed nozzle (ϵ = 172:1) and return. The coolant flows are merged, and 6% of the total engine hydrogen flow is used to bypass both turbines to provide cycle power balance margin and thrust control. The remaining hydrogen flow first drives the fuel pump turbine, and then drives the oxidizer pump turbine. After driving the oxidizer pump turbine, a small amount of heated hydrogen is tapped off for hydrogen tank pressurization. The remaining hydrogen flow is then injected into the combustion chamber.

TABLE I

OTV ENGINE REQUIREMENTS

- Thrust: 15,000 lb (Nominal)
- Propellants: Hydrogen and Oxygen
- Technology Base: 1980 State-of-the-art
- Engine Mixture Ratio: Nominal = 6.0 Range = 6.0 to 7.0
- Propellant Inlet Conditions:

		H_2	O_2
Boost Pump	NPSH, ft	15	2
	Temp. °R	37.8	162.7

- Service Lift Between Overhauls: 300 cycles and 10 hrs
- Service Free Life: 60 cycles or 2 hrs
- Engine Nozzle: Contoured Bell with Extendible/Retractable Section
- Maximum Engine Length with Nozzle Retracted:
  Nominal = 60"
- Gimbal Angle: +15°, -6° Pitch
  ±6° Yaw
- Provide Gaseous Hydrogen and Oxygen Tank Pressurization
- Man-Rated: Capable of providing abort return to the Space Shuttle Orbiter
- Meet Orbiter Safety and Environmental Criteria

TABLE II

ADVANCED EXPANDER CYCLE ENGINE OPERATING SPECIFICATION

SERIES TURBINE DRIVE CYCLE

Rated Vacuum Thrust = 15,000 lb
Stowed Length = 60 in.

	Engine Mixture Ratio 6.0	Engine Mixture Ratio 7.0
Engine		
Vacuum Thrust, lb	15,000	15,000
Vacuum Specific Impulse, sec.	475.4	471.0
Total Flow Rate, lb/sec	31.56	31.85
Mixture Ratio	6.0	7.0
Oxygen Flow Rate, lb/sec	27.05	27.87
Hydrogen Flow Rate, lb/sec	4.51	3.98
Thrust Chamber		
Vacuum Thrust, lb	15,000	15,000
Vacuum Specific Impulse, sec	475.4	471.0
Chamber Pressure, psia	1,200	1,162
Nozzle Area Ratio	435	435
Mixture Ratio	6.0	7.0
Throat Diameter, in.	2.79	2.79
Chamber Diameter, in.	5.34	5.34
Chamber Length, in.	18.0	18.0
Chamber Contraction Ratio	3.66	3.66
Nozzle Exit Diameter, in.	58.2	58.2
Percent Bell Nozzle Length	81.8	81.8
Nozzle Length, in.	84.4	84.4
Combustion Chamber Coolant Flow Rate, lb/sec	3.834	3.383
Slotted Copper Chamber Area Ratio	10.6	10.6
Chamber Pressure Drop, psia	92	76
Coolant Inlet Temperature, °R	90	90
Chamber Coolant Temperature Rise, °R	411	431
Fixed Tube Bundle Nozzle Flow Rate, lb/sec	0.677	0.597
Tube Bundle Nozzle Area Ratio	172	172
Tube Bundle Coolant Pressure Drop, psia	10	8
Tube Bundle Coolant Temperature Rise, °R	640	672

TABLE II

OPERATING SPECIFICATION (cont.)

	ENGINE MIXTURE RATIO 6.0		ENGINE MIXTURE RATIO 7.0	
Boost Pumps	LOX	LH_2	LOX	LH_2
Inlet Flow, GPM	171	456	175	401
Inlet Pressure, psia	16	18.5	16	18.5
Vapor Pressure, psia	15.2	18	15.2	18
Inlet Temperature, °R	167.2	37.8	162.7	37.8
NPSH, ft (not including TSH)	2	15	2	15
Discharge Pressure, psia	57	50	57	50
Head Rise, ft	82.3	1026	82.3	1026
Speed, RPM	7400	29650	7400	29650
Suction Specific Speed, $(RPM)(GPM)^{1/2}/ft^{3/4}$	25560	43080	25860	40400
Specific Speed, $(RPM)(GPM)^{1/2}/ft^{3/4}$	3540	3500	3580	3275
Efficiency, %	66	78	66	77
Boost Pump Hydraulic Turbines				
Flow (GPM)	16.7	129	17.3	116
Efficiency, %	52	66	52	66
Horsepower	6.1	10.7	6.3	9.6

TABLE II

OPERATING SPECIFICATION (cont.)

Main Pumps	ENGINE MIXTURE RATIO 6.0		ENGINE MIXTURE RATIO 7.0	
	LOX	LH_2	LOX	LH_2
Inducer				
Flow, GPM	194	547	199	481
Head Rise, ft	460	5080	450	4645
Efficiency,%	75	82	75	81
Horsepower	34	64.2	34	49.4
Inlet Pressure, psia	48	49	48	49
NPSH, ft	66	997	66	997
Suction Specific Speed, $(RPM)(GPM)^{1/2}/ft^{3/4}$	20880	11860	21000	11100
Specific Speed, $(RPM)(GPM)^{1/2}ft/^{3/4}$	4870	3500	4980	3500
Stage 1				
Flow, GPM	223	572	229	503
Head Rise, ft	2450	26870	2410	24640
Efficiency, %	71	71	71	70
Horsepower	219	386	221	317
Specific Speed, $(RPM)(GPM)^{1/2}/ft^{3/4}$	1490	1025	1520	1024
Stages 2 & 3				
Flow, GPM	---	478	---	421
Head Rise, ft	---	23840	---	21940
Efficiency, %	---	71	---	70
Horsepower	---	286	---	236
Specific Speed, $(RPM)(GPM)^{1/2}/ft^{3/4}$	---	1025	---	1022

TABLE II

OPERATING SPECIFICATION (cont.)

	ENGINE MIXTURE RATIO 6.0		ENGINE MIXTURE RATIO 7.0	
	LOX	LH_2	LOX	LH_2
Overall				
Head Rise, ft	2910	79630	2860	73165
Efficiency, %	61.5	63.3	61.5	62.3
Horsepower	253	1030	255	838
Speed, RPM	34720	90000	34470	89800
Main Pump Turbines				
Inlet Pressure, psia	1512	2344	1469	2176
Inlet Temperature, °R	489	535	514.6	557
Flowrate lb/sec	4.22	4.22	3.71	3.71
Gas Properties				
Cp, Specific Heat at Constant Pressure, BTU/lb-°R	3.652	3.652	3.652	3.652
γ, Ratio of Specific Heats	1.395	1.395	1.395	1.395
Shaft Horsepower	253	1030	255	838
Pressure Ratio (Total to Static)	1.14	1.540	1.153	1.471
Static Exit Pressure, psia	1326	1522	1274	1479
Static Exit Temperature, R°	471.6	473.5	494.3	499.3
Efficiency, %	66.7	76.8	65.7	75.8
Turbine Bypass Flowrate lb/sec	0.27	0.27	0.24	0.24

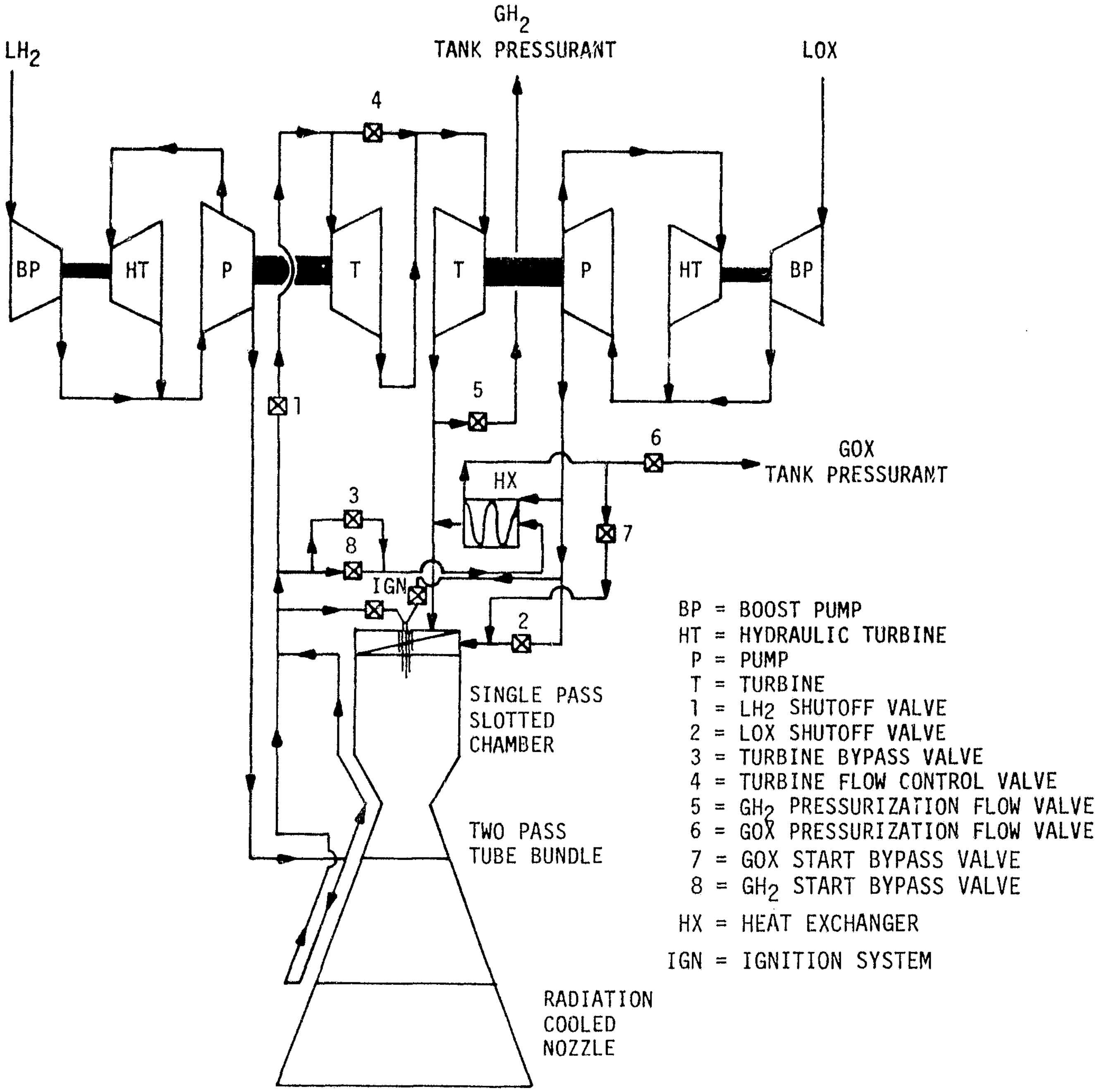

Figure 1. Advanced Expander Cycle Engine Flow Schematic

At rated thrust operation, oxidizer flows from the main pump discharge directly to the thrust chamber and is injected in a liquid state. A small amount of oxidizer is tapped off and heated by the hydrogen turbine bypass flowrate in a heat exchanger to provide LOX tank pressurization.

A lightweight, state-of-the-art Columbium nozzle extension was selected as the baseline on the basis of experience gained on the Transtage, Apollo, SPS, and OMS engine programs.

The engine is also capable of operating in a tank-head idle mode and is adaptable to extended low-thrust operation at thrust level of 1.5K lb.

The purpose of the tank-head idle mode is to thermally condition the engine without non-propulsive dumping of propellants. This is a pressure-fed mode of operation at a thrust level of approximately 50 lbs with a vacuum specific impulse estimated to be 400 secs. During this mode of operation, the main fuel and oxygen valves (numbers 1 and 2 on the schematic) are closed. All of the fuel bypasses the turbines through valve number 3 so that the pumps are not rotating. The heat exchanger in the turbine bypass line gasifies the oxygen which then flows through valve number 7 to the chamber. Tank pressurization is not supplied during this operating mode and valves 5 and 6 are closed. The pressurization valves are opened as the engine is brought up to steady-state, full thrust operation.

The OTV point design engine is adaptable to operation at 10% of rated thrust (i.e., 1.5K lbF) with minor modifications. This low-thrust operating point is a dedicated condition and the engine is not required to operate at both the 15K and 1.5K thrust levels on the same mission. To operate at low thrust, the oxidizer injection elements must be changed to one of smaller size and an orifice must be installed in the line downstream of the chamber coolant jacket. Performance during this mode of operation is presented in Section V.

The engine pressure schedule at rated thrust operation is shown on Table III for both the nominal and off-design mixture ratio conditions. The chamber pressure of 1200 psia was selected on the basis of cycle optimization and trade-off studies which evaluated specific impulse and weight changes with chamber pressure. Further optimization and trade-offs are planned in future work and some changes in operating chamber pressure and performance are anticipated.

TABLE III

ADVANCED EXPANDER CYCLE ENGINE PRESSURE SCHEDULE
SERIES TURBINES DRIVE CYCLE
F = 15,000 lb

	ENGINE MIXTURE RATIO 6.0		ENGINE MIXTURE RATIO 7.0	
	LOX	LH_2	LOX	LH_2
Pressure, psia(1)				
Boost Pump Inlet	16	18.5	16	18.5
Boost Pump Discharge	56	50	56	50
Main Pump Inlet	48	49	48	49
Main Pump Discharge	1487	2531	1463	2328
ΔP Line	25	10	26	8
Main Shutoff Valve Inlet	1462	2521	1437	2320
ΔP Shutoff Valve	15	25	16	19
Shutoff Valve Outlet	1447	2496	1421	2301
ΔP Line	15	30	16	23
Coolant Jacket Inlet	---	2466	---	2278
ΔP Chamber Coolant Jacket	---	92	---	76
Coolant Jacket Outlet	---	2374	---	2202
ΔP Line	---	30	---	26
Fuel Turbine Inlet	---	2344	---	2176
Fuel Turbine Pressure Ratio (Total/Static)	---	1.540	---	1.471
Fuel Turbine Static Exit	---	1522	---	1479
Fuel Turbine Total Exit	---	1560	---	1516
ΔP Warm Gas Duct	---	48	---	47
OX Turbine Inlet	---	1512	---	1469
OX Turbine Pressure Ratio (Total/Static)	---	1.14	---	1.153
OX Turbine Static Exit	---	1326	---	1274
OX Turbine Total Exit	---	1360	---	1307
ΔP Warm Gas Duct	---	34	---	33
Injector Inlet	1434	1326	1405	1274
ΔP Injector	215	109	227	96
Injector Face	1217	1217	1178	1178
Chamber	1200	1200	1162	1162

(1) All pressures are total pressure except where noted.

IV. ENGINE LAYOUT AND COMPONENT DESIGN DRAWINGS

This section presents the Advanced Expander Cycle Engine layout drawing and accompanying major component drawings. These are preliminary designs which provide adequate design detail for determining component weights but are not adequate for manufacturing purposes and require further design iterations. A summary description of the components is also provided. These descriptions are brief and further design information is available in the final report for this contract.

A. ENGINE LAYOUT

The engine assembly layout drawing is presented on Figure 2 which shows the engine top and side views. The engine is 60 in. long with the extendible nozzle in the stowed position. This length is measured from the top of the gimbal block to the end of the tube bundle nozzle. The engine is 109.6 inches long with the extendible nozzle deployed. Approximately 10.4 inches of potentially available deployed length is lost in the area of the extendible nozzle deployment mechanism and attachment plane. Further design refinements could increase the deployed length to a maximum of 120 in. with a resulting area ratio of 473:1 and a performance increase of 1.8 sec as compared to Table II.

B. GIMBAL ASSEMBLY

The gimbal assembly (Figure 3) consists of two major subassemblies: (1) thrust mount and (2) monoball, thrust lug and bolt assembly and anti-rotation tie rod assembly. The structure provides the monoball gimballing capability and the attach points for the hydraulic actuators and TCA. Thrust is transmitted from the back of the injector flange through the struts of the thrust mount and monoball assembly to the propellant tank bottom flange.

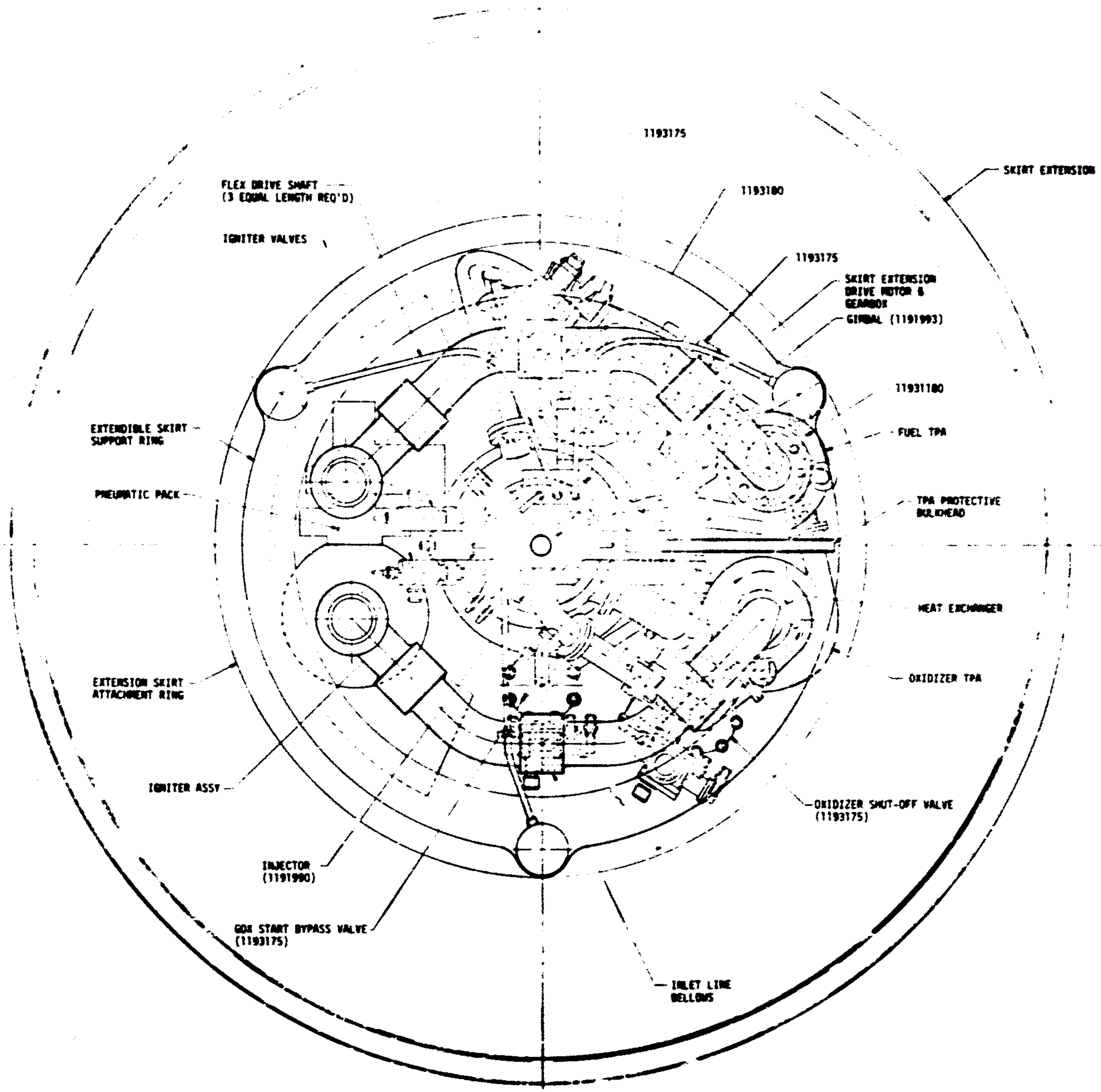

Figure 2. Engine Layout (ALRC Dwg. #1193100) (Sheet 1 of 3)

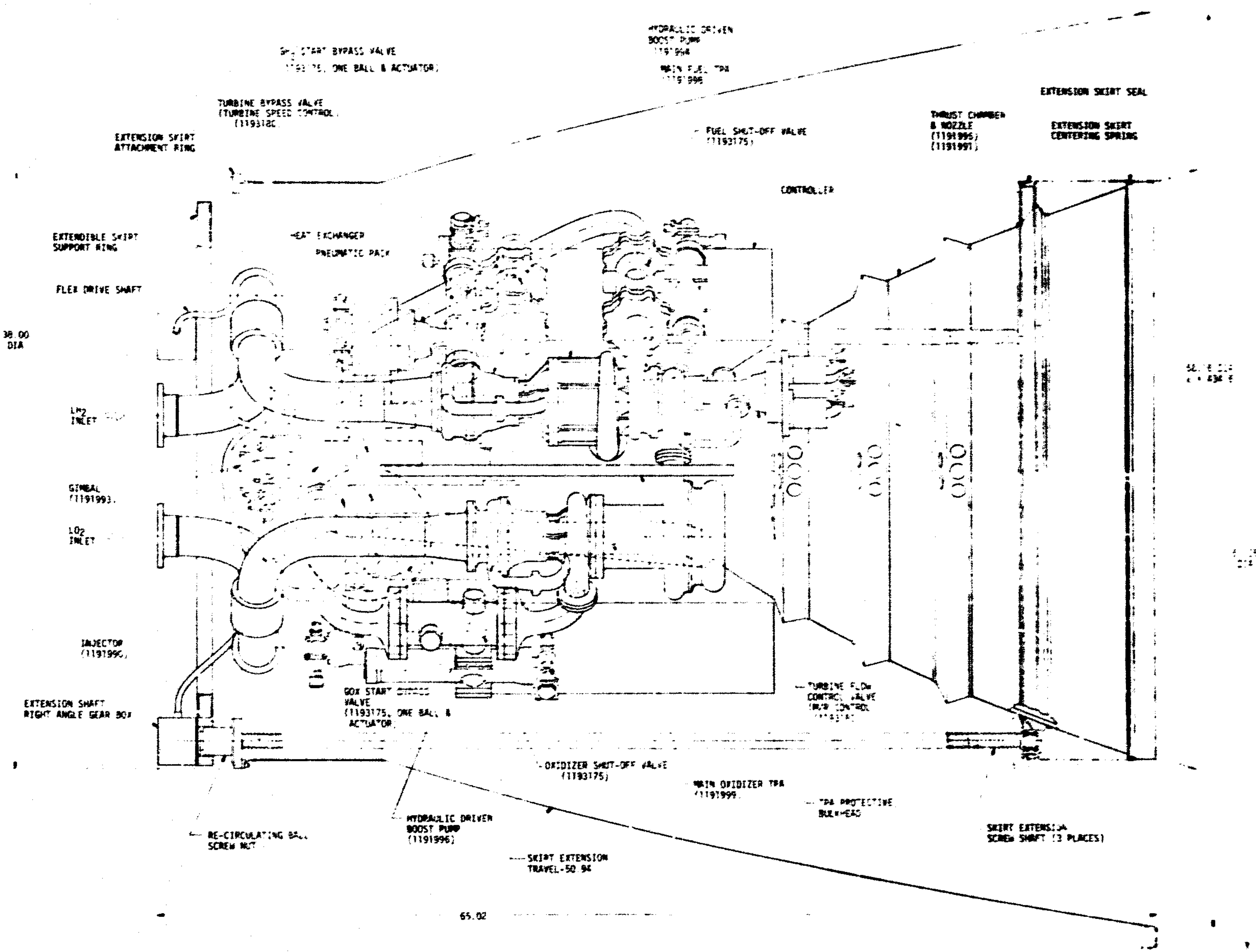

Figure 2. Engine Layout (ALRC Dwg. #1193100) (Sheet 2 of 3)

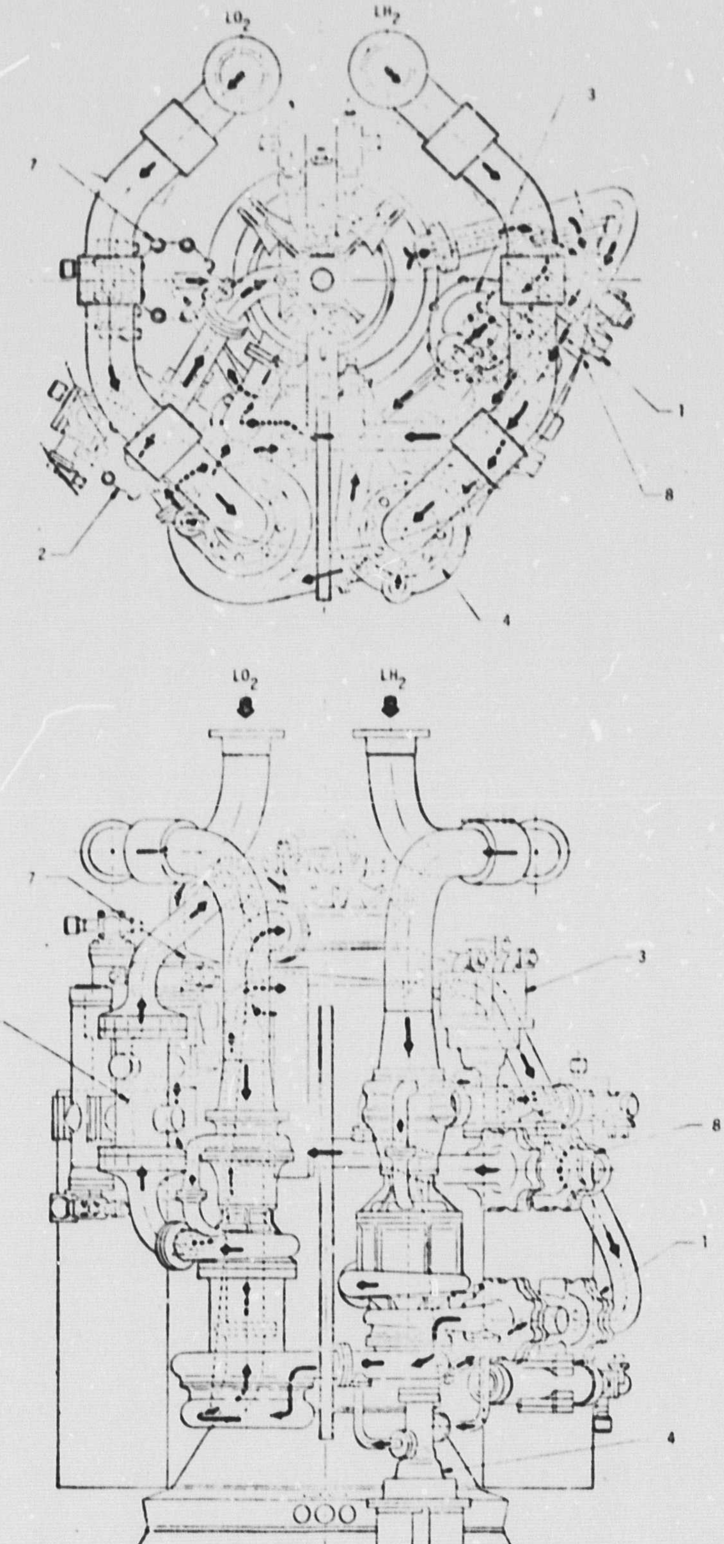

Figure 2. Engine Layout (ALRC Dwg. #1193100) (Sheet 3 of 3)

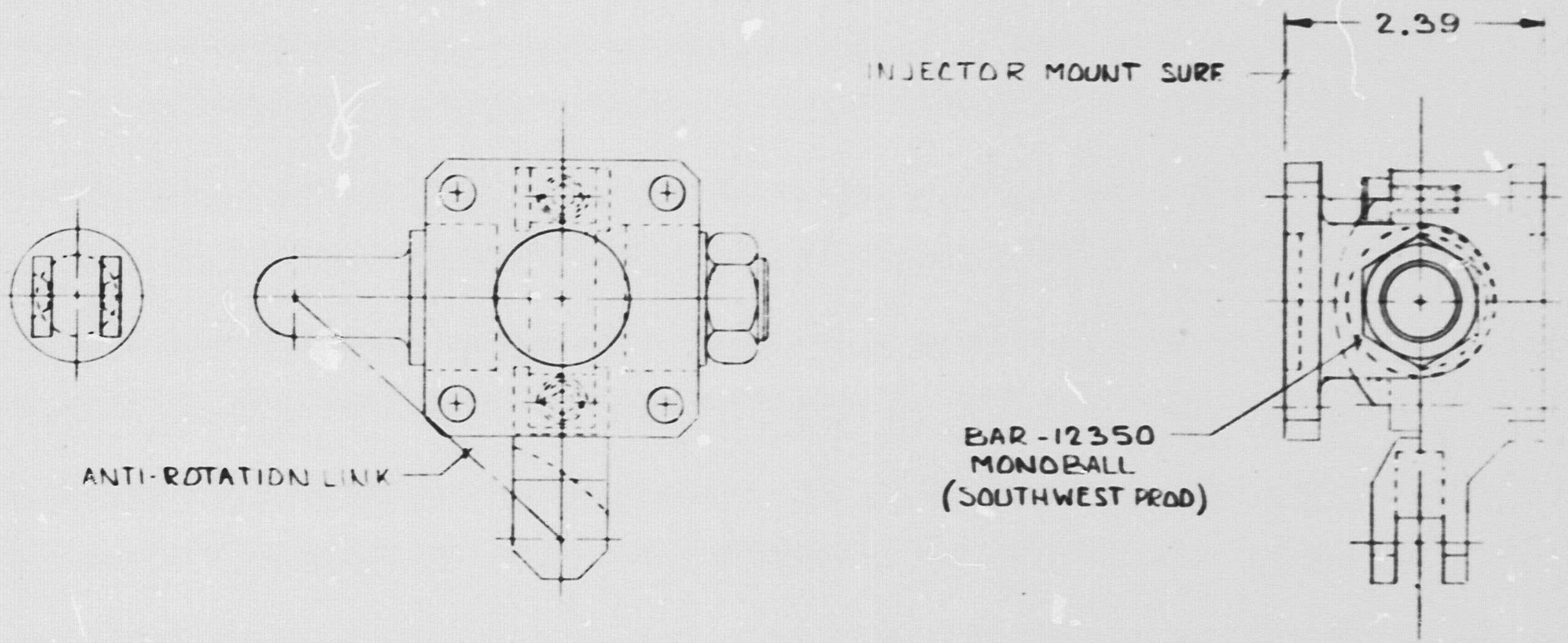

Figure 3. Gimbal Assembly (ALRC Dwg. #1191993)

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C. IGNITER/INJECTOR ASSEMBLY

The OTV ignition system employs redundant igniters as shown mounted on the OTV injector (see Figure 4). Each igniter is a small thruster which can accept either liquidous, two phase, or gaseous propellants. These are ignited by using a very low energy spark. The igniter produces a hot gas torch of sufficient energy to provide reliable rapid main stage ignition.

The ignition system consists of 5 major components: (1) a GLA spark plug, (2) Model 427200-4871 Valcor coaxial type poppet valves; (3) a stainless steel/nickel body which forms or contains all manifolding and seals, propellant metering and injection orifices, a platform for mounting the spark plug and valves plus all necessary instrumentation ports and a flange for attachment to the injector, (4) a hydrogen cooled chamber, and (5) a high voltage GLA capacitance discharge power supply.

The capacitance discharge power supply is integral with the spark plug. The igniter assembly is located in the injectors oxidizer manifold cover plate directly below the gimbal mount surface. The igniter propellant valves are located in close proximity to the igniter/injector as shown on the engine layout drawing (see Figure 2).

A layout of the injector configuration is shown in Figure 4. The injector uses coaxial elements because this element type has an extensive history of operation with GH_2/LO_2 propellants over a broad range of thrusts and chamber pressure. The injection pattern is a four row array of 84 elements uniformly distributed over the injector face. Since the hydrogen is injected as a gas, and the cryogenic LOX immediately flashes into a gas upon injection into the hot combustion chamber, combustion is expected to occur very close to the injector face. This results in the need for injector face cooling. Regenerative cooling of the injector face, coupled with

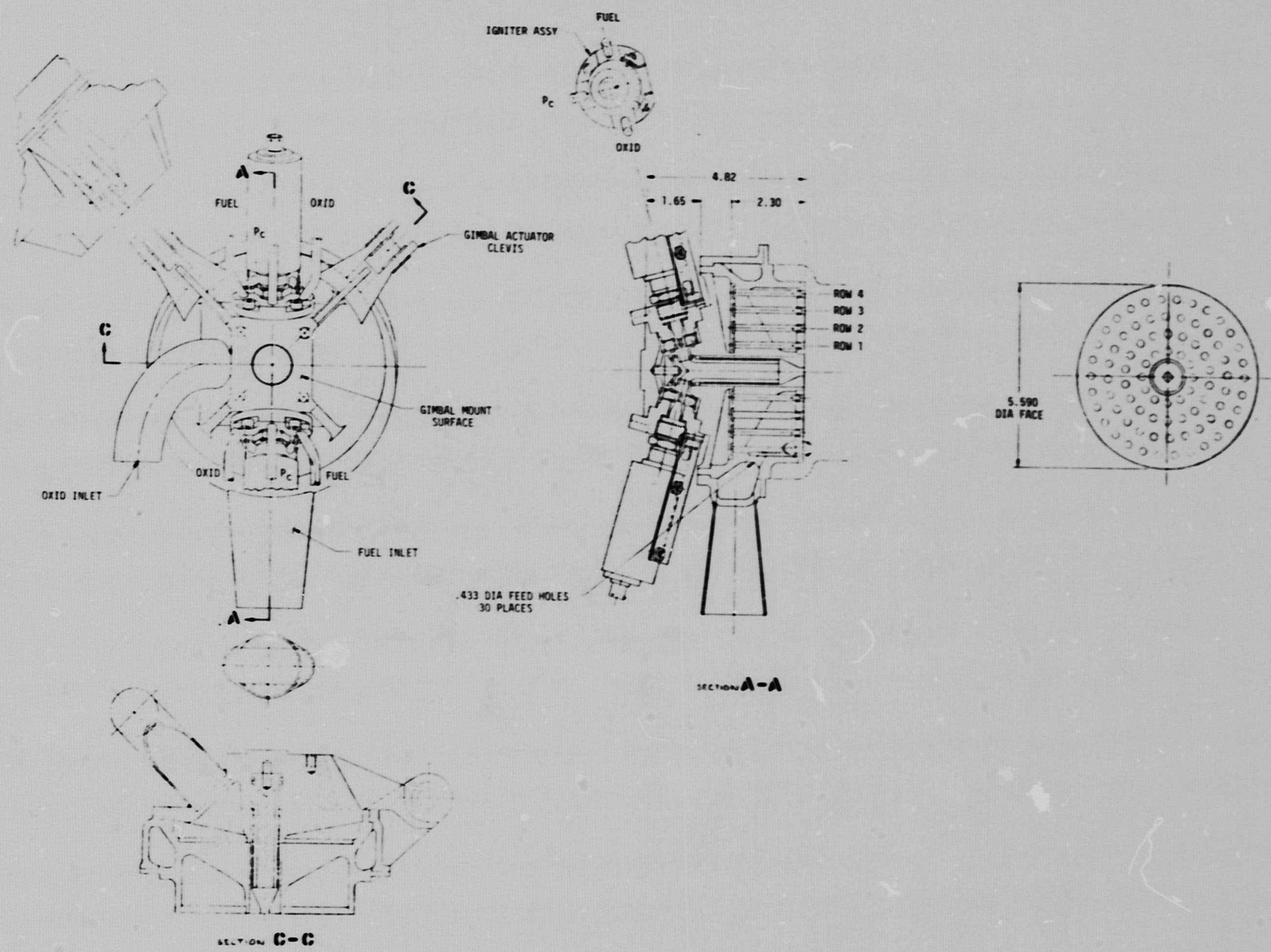

Figure 4. Igniter/Injector Assembly (ALRC Dwg. #1191990)

discreet face fuel film cooling provide the most reliable face cooling. The injector face plate material is a laminate of photoetched copper face platelets brazed to a structural steel back up plate. The extreme., accurate photoetched flow control passages assure uniform flow across the entire injector face. The photoetched face platelet concept also permits the incorporation of a filter screen into the platelet stack. The face plate concept precludes problems associated with flow control and flow distribution encountered with Rigimesh, a stainless steel wire material used on a variety of rocket engines with varied success.

The fuel inlet torus surrounding the injector body is designed for constant flow velocity to assure uniform fuel distribution into the injector fuel manifold through 30 equally spaced holes. The oxidizer manifold which is located forward of the fuel manifold directly below the redundant torch igniters is designed for uniform oxidizer distribution to the 84 injection elements. The oxidizer tubes which are recessed approximately one tube diameter into the face plate are held concentrically within the fuel discharge orifice by four small tabs integral with the faceplate.

Although combustion instability problems are not expected, a resonator cavity is provided around the injector periphery. The cavity is designed so that dynamic pressure oscill ions do not exceed $\pm$ 5%.

D. CHAMBER AND TUBE BUNDLE NOZZLE

A design layout for the combustion chamber is illustrated in Figure 5. The chamber ID is 5.34 in., is cylindrical for 13.54 in. and then converges at a 30° half angle to a throat diameter of 2.79. The thrust chamber gas side wall contains 113 coolant slots which are equally spaced in a zirconium copper liner. The coolant slots are closed out with

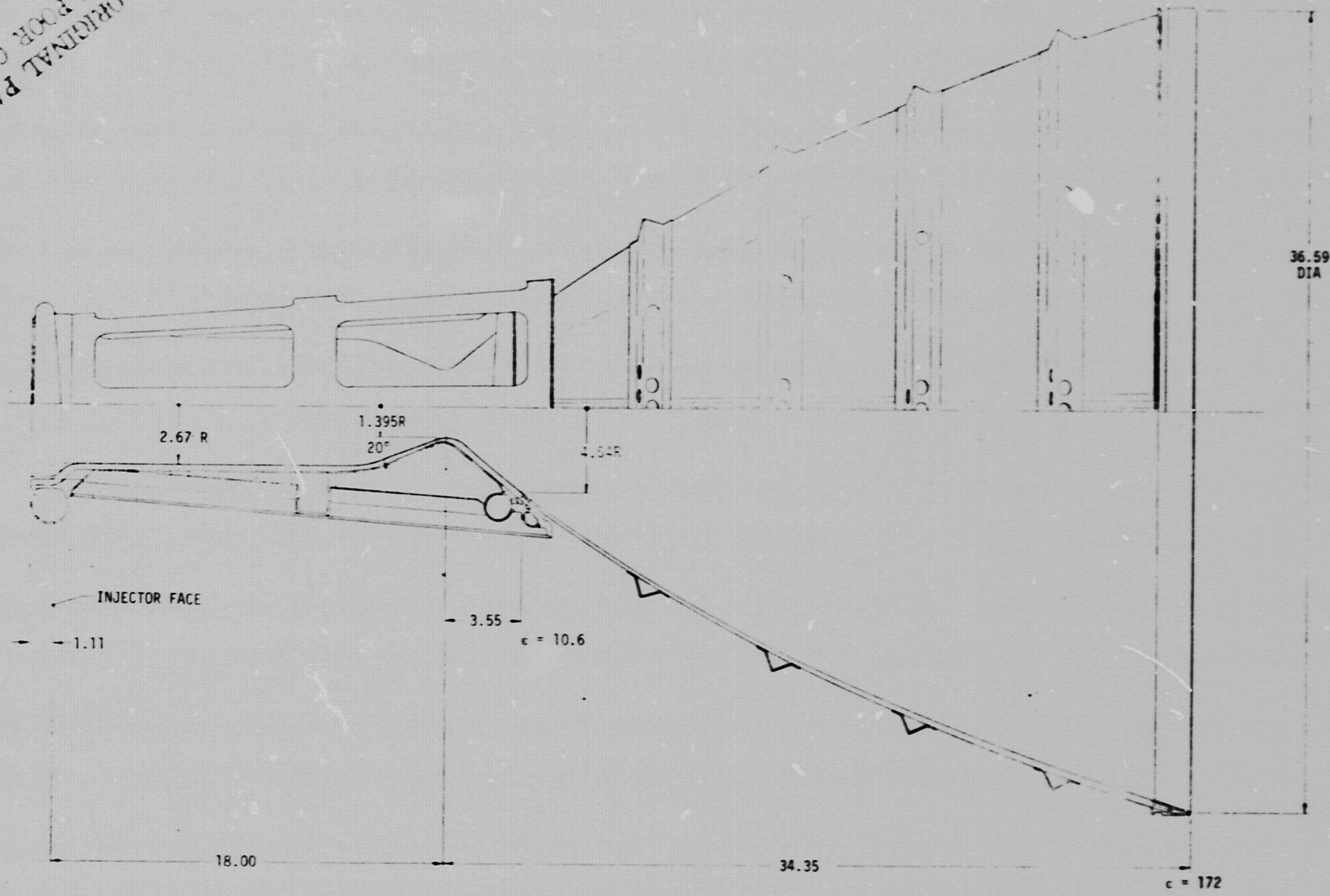

Figure 5. Chamber and Tube Bundle Nozzle (ALRC Dwg. #1191991)

electroformed nickel. The coolant enters the chamber at an area ratio of 10.6. The inlet manifold is located 4.20 in. below the throat. The coolant flows axially for a distance of 22.20 in. toward the forward end of the chamber. The coolant is collected in a manifold outboard of the resonator cavities and exits radially into the chamber's coolant outlet manifold.

The selection of thrust chamber geometry (contraction ratio and combustor length) is influenced by engine cycle considerations. Studies show that the total heat load (coolant temperature rise) is increased as L' and contraction ratio decreases. This increases the turbine inlet temperature but increases system pressure drops. Optimization studies resulted in the selection of a 3.66 contraction ratio and a combustor length L' of 18 in.

Twelve resonator cavities are located at the forward end of the thrust chamber and are bounded by the injector body outside diameter and chamber inside diameter. The coolant passages in line with the 12 partitions between the resonator cavities will extend partially through the partition to regeneratively cool the partition.

A conical support structure surrounds the entire chamber to provide a load path for gimbal induced loads from the nozzle extension as well as to provide a means of attachment for engine components. The conical support structure has a circumferential channel located at its inside diameter approximately midway along its length to provide additional structural rigidity. Openings in the cone provide access for bolting components and routing propellant lines.

The regenerative cooled tube bundle nozzle is physically attached to the chamber as shown in Figure 5. A two pass tube bundle configuration was selected.

The forward end of the nozzle is located at an area ratio of 10.6:1, which is at a contour ID of 9.100 in. The tube bundle nozzle ends at an area ratio of 172:1 which is 34.35 in. below the throat at a contour ID of 36.59 in.

A total of 326 coolant tubes are spaced equally around the nozzle contour. The tubes taper from 0.089 in. O.D. with a wall thickness of 0.007 in. at the forward end to 0.361 in. O.D. with a wall thickness of 0.010 in. at the aft end. A-286 has been preliminarily selected as the tube bundle material in this design although further study is required.

To assure tube bundle rigidity, four circumferential ring stiffeners are brazed to the exterior of the tube bundle. To assure that the deployable radiation cooled nozzle will properly align, center itself, and seal on the regenerative cooled nozzle, a stainless steel flange has been designed to be brazed to the tube bundle. This flange is located approximately eight inches from the aft end as shown in Figure 2. This flange accommodates the bushings for the extension/retraction mechanism screw, a gasket seal gland and one of two spring loaded alignment rings. The second alignment ring is located in the outer periphery of the copper turn around manifold.

E. RADIATION COOLED NOZZLE EXTENSION AND DEPLOYMENT MECHANISM

The radiation cooled nozzle extension assembly is shown in Figure 2 (engine layout drawing). It consists of a contoured radiation cooled nozzle, an extension/retraction mechanism and a mechanical drive system.

The radiation cooled nozzle, in the extended position, is located 34.35 in. below the throat at a contour ID of 36.59 in. and extends to an exit area ratio of 435:1. The retracted position of the nozzle is such

that its exit plane is at the same axial station as that of the regeneratively cooled nozzle (172:1) which is 34.35 in. below the throat.

The radiation cooled nozzle extension that is exposed to the hot gases is 49.6 in. long. A thin wall cylindrical ring assembly, approximately 10.4 in. long, containing the nozzle attachment flange is an integral part of the nozzle. This ring assembly is not exposed to the hot products of combustion. Its function is to permit extension of the nozzle flange up to the deployment ring where it is bolted to the extension/retraction mechanism. The radiation cooled nozzle incorporates a bolt-on flange and can be removed and replaced from the aft end.

Design of the radiation cooled nozzle is based upon the OMS engine nozzle design. The nozzle material is C 103 columbium alloy. An oxidation resistant coating (an aluminide or silicide slurry) is required on all surfaces of the nozzle.

Redundant seals are employed between the extendible nozzle and the regeneratively cooled nozzle to prevent hot gas leakage. Preliminary studies indicate that the low temperatures on the backside tip of the regeneratively cooled nozzle will permit use of an elastomeric type diametric seal. The nozzle joint is also sealed by an elastomeric gasket type face seal. A positive seal is assured by serrations machined in the surfaces of all parts that contact the seal. The amount of gasket "crush" between the mating parts is controlled by interlocking lips. This sealing concept is both positive and insensitive to machining tolerances. It does require that the seal be held in compression during the engine firing mode.

The extendible nozzle is aligned directly with the fixed nozzle. A spring-loaded nozzle centering device depicted in Figure 2 consists of a one piece spring positioned in a groove on the lower support ring for the

extension/retraction mechanism. A second centering spring is positioned just upstream of the diametric seal located on the tip of the regeneratively cooled nozzle. This spring assures final centering of the nozzle and a more uniform loading on the seal.

The guide and support system concept consists of three tubular steel screw shafts located axially 120° apart around the fixed portion of the nozzle. Each shaft is mounted in sleeve bearings located in the lower tube bundle "V" band support. The other end of the threaded shaft is mounted in a bearing in the gearbox which is rigidly attached to the upper support ring, see Figure 2.

A small, reversible 28 volt DC drive motor, suitable for vacuum operation, having integral spur reduction gear sets and provisions for three separate power take-offs is mounted on the engine structure just below the gimbal plane. A flexible drive shaft transmits power to each screw shaft. The motor contains a mechanical lock that is automatically activated to prevent movement of the drive train components whenever the nozzle is in the extended or retracted position. The motor also contains a tool attachment to manually extend or retract the nozzle if necessary.

F. LH_2 BOOST PUMP

The LH_2 boost pump is illustrated in Figure 6. It is driven by a partial admission hydraulic tip turbine. The turbine is driven in turn by hydrogen flow obtained from the first stage of the main hydrogen turbopump. The hydrogen leaving the tip turbine is mixed with the boost pump through-flow and then reenters the main hydrogen pump. The boost pump discharge flow is axial to facilitate close coupling of the boost and main turbopump assemblies in the engine package.

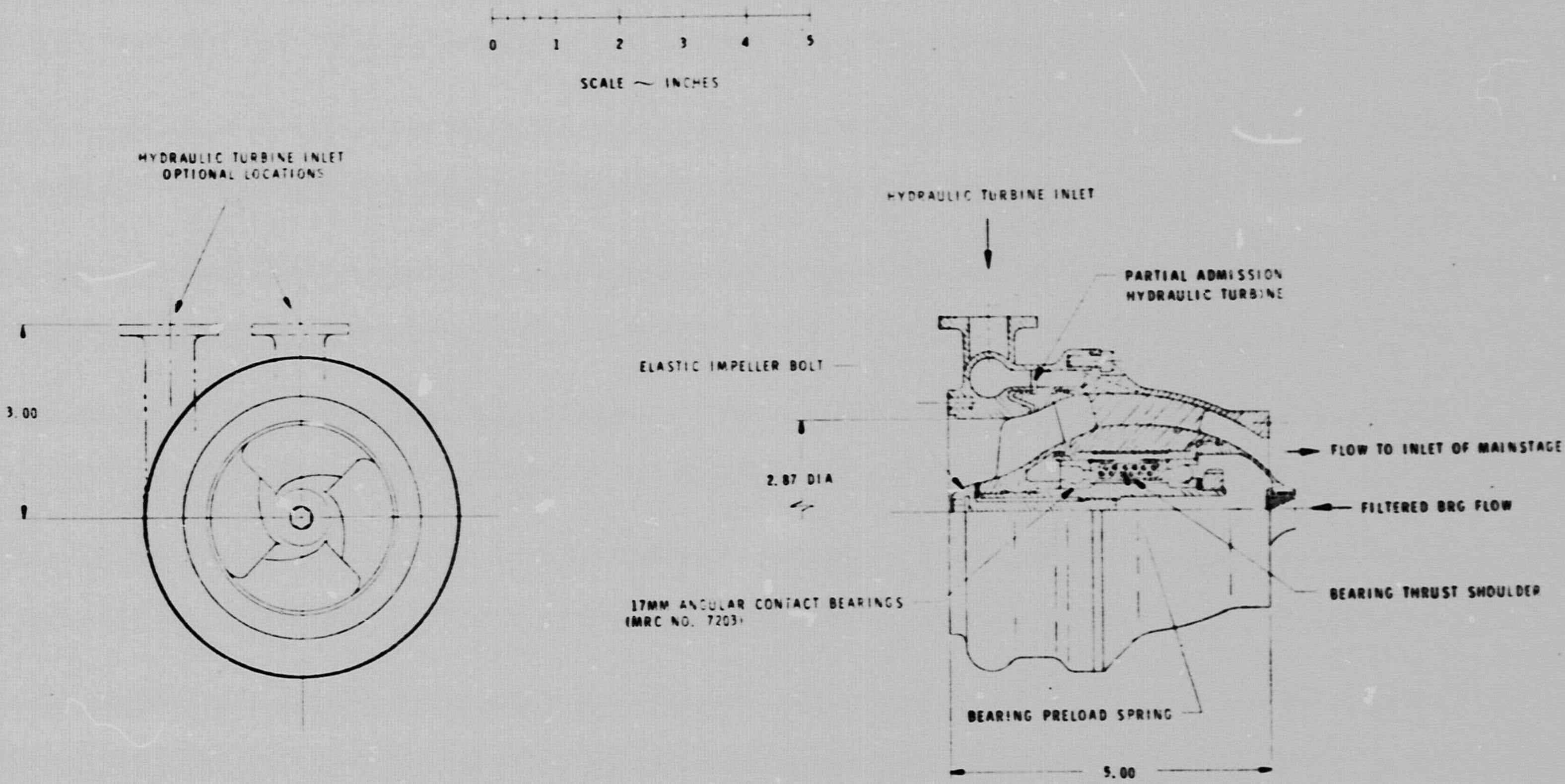

Figure 6. LH_2 Boost Pump (ALRC Dwg. #1191994)

The hydraulic turbine is hub mounted and yields an efficiency of 66% with a head change of 7136 feet with a flow of 129 GPM.

The boost pump operates at an NPSH of 15 ft, delivers a discharge pressure of 50 psia and is 78% efficient.

G. LO_2 BOOST PUMP

The low speed LO_2 boost pump is illustrated in Figure 7. Like the LH_2 boost pump it is also driven by a partial admission hydraulic tip turbine. The turbine is driven by LO_2 flow from the main pump discharge. The LO_2 boost pump exit flow is axial to facilitate close coupling of the boost and main turbopump assemblies in the engine package.

The NPSH of the boost pump is 2 ft and the discharge pressure is 57 psia. Boost pump efficiency is 66%.

The hydraulic turbine has an efficiency of 52% and is driven by 16.7 GPM of flow.

H. LH_2 TPA

The main high-speed hydrogen turbopump is shown on Figure 8 and its operating specification is shown on Table II. The pump is a three stage machine that is driven by a two-stage turbine. The pump discharge pressure is 2531 psia and operates at a speed of 90,000 RPM. The turbine inlet temperature is 535°R which is a benign operating environment and desirable for this man-rated application.

The pump and turbine manifold are a one piece housing with a flange joint at the pump impeller and turbine exhaust ends. The pump impeller and

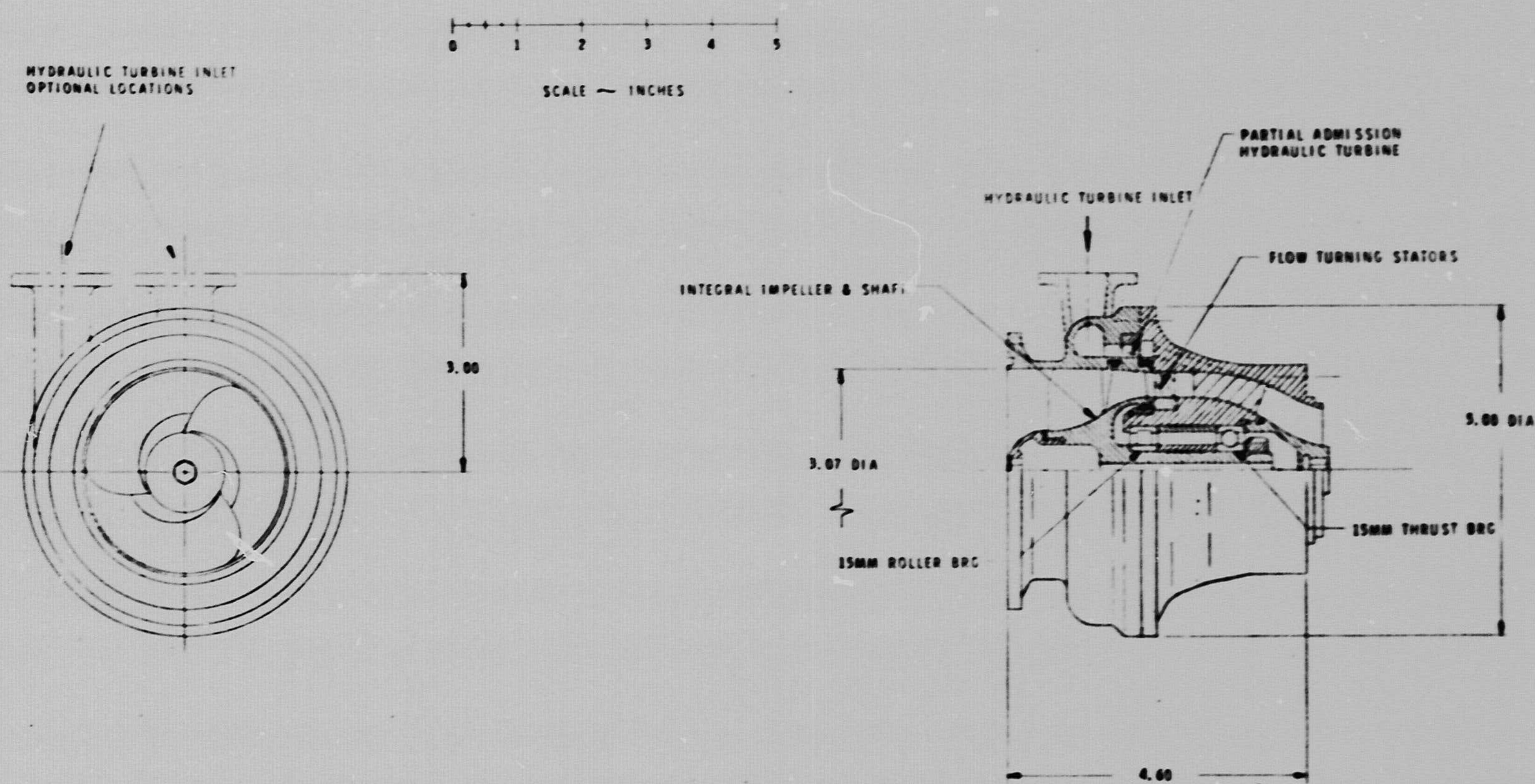

Figure 7. LO_2 Boost Pump (ALRC Dwg. #1191996)

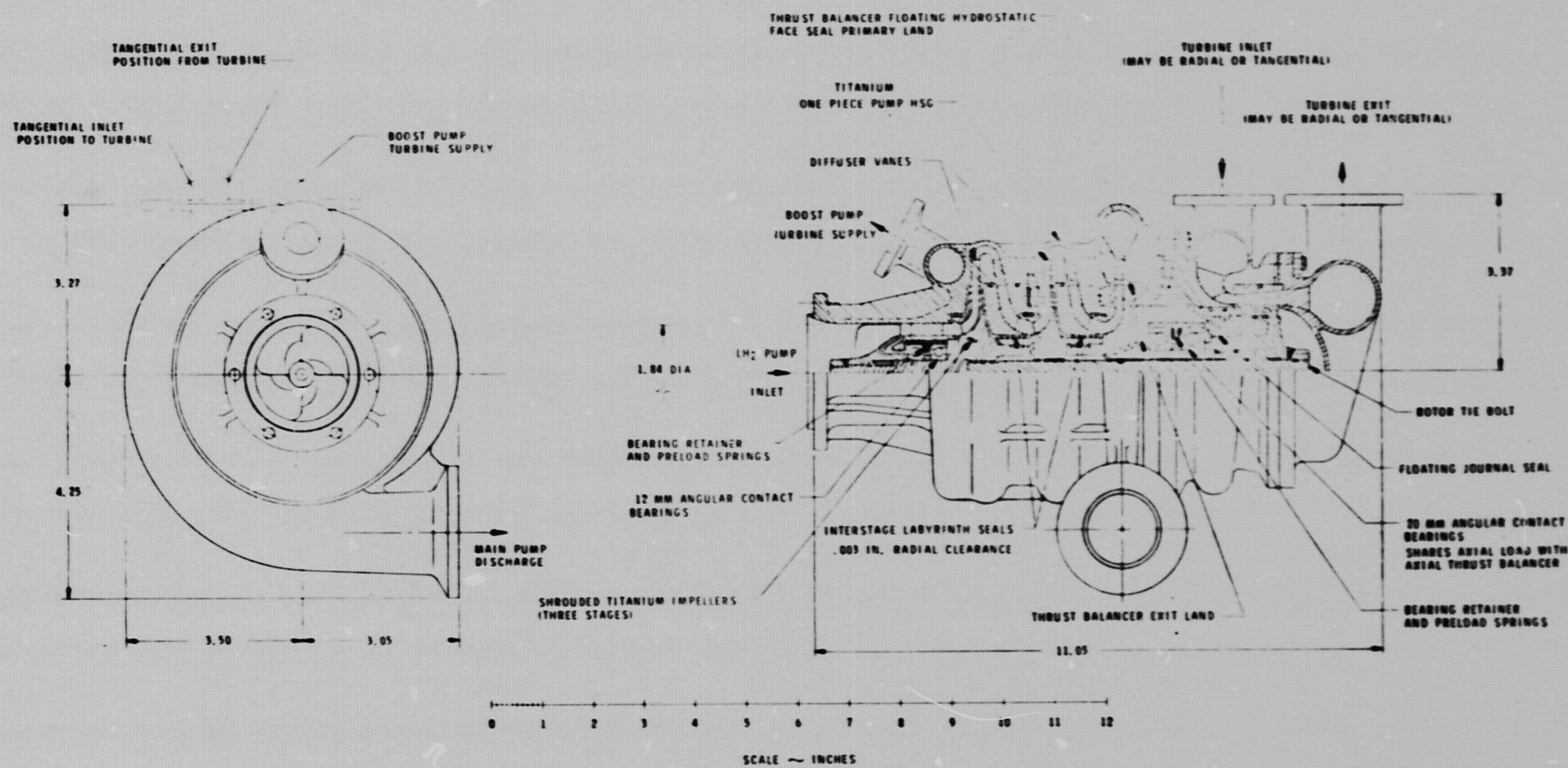

Figure 8. LH_2 TPA (ALRC Dwg. #1191997)

inducer is made of titanium. The pump diffuser vanes and crossover passages are encased in removable disks. The rotating assembly is a built-up construction with an elastic tie bolt to maintain structural integrity. The rotating assembly is supported on a set of angular contact bearings between the inducer and first impeller and between the third impeller and the first turbine disk. The pump end-bearing set supports radial loads but permits axial motion. The turbine bearing set provides radial support and axial restraint. A hydraulic thrust balancer is located on the backside of the third impeller. Pressure from the third stage impeller operates the balancer and the exit flow returns to the second impeller inlet. To prevent turbine gas from entering the bearing cavity, a high pressure turbine seal controls leakage to the turbine and an adjacent seal controls flow to the bearings.

The pump impellers are shrouded with straight labyrinths at the front shroud impeller inlet and on the shaft between each impeller. The front of all three impellers is identical. The back of the third stage differs from the back of the first and second stage by the axial thrust balancer.

At the turbine, a double floating ring seal is used to control liquid hydrogen leakage into the gaseous hydrogen of the turbine and liquid hydrogen to the bearing. To keep a positive flow to the turbine and prevent backflow of turbine gases, the seals are pressurized from the third stage impeller.

A pair of angular contact bearings, preloaded back to back, are used to maintain a minimum stiffness and avoid skidding of the rolling elements. The pump end bearings are 10mm inside diameter selected for hydraulic passage clearance. The turbine end bearings are 20mm inside

diameter and are not restricted at the outside diameter but are maximized at the base to transmit the torque and provide for a high stiffness shaft. The pump end bearings provide radial support but are free to move axially in the retainer. The turbine end-bearings provide radial support and share the axial load with the hydraulic thrust balancer.

I. LO_2 TPA

The main high-speed oxygen turbopump assembly is shown on Figure 9 and its operating specification on Table II. The pump is a single stage machine that is driven by a single stage turbine.

The main pump is made up of two pumping elements, an inducer and an impeller, directly connected together. The pump is suction specific speed limited (20,880 RPM x $GPM^{1/2}/ft^{3/4}$), has a discharge pressure of 1487 psia and runs at a speed of 34,720 RPM.

The inducer, which can run at a relative high speed without cavitating, provides sufficient head to the impeller to keep it from cavitating. The head split is approximately 15% to 85%.

The inducer has a cylindrical tip so that it is not sensitive to axial displacement. The inducer receives flow directly from the boost pump at a pressure of 48 psia. The impeller has front and back shrouds with cylindrical labyrinths. Flow from the bearing package is returned between the inducer and the impeller where the pressure is high enough to prevent flashing of the liquid oxygen due to the heat removed from the bearings.

The seal package between the oxygen in the pump and the heated hydrogen in the turbine is a conventional design using a helium purge. This seal package consists of five circumferential type shaft seals.

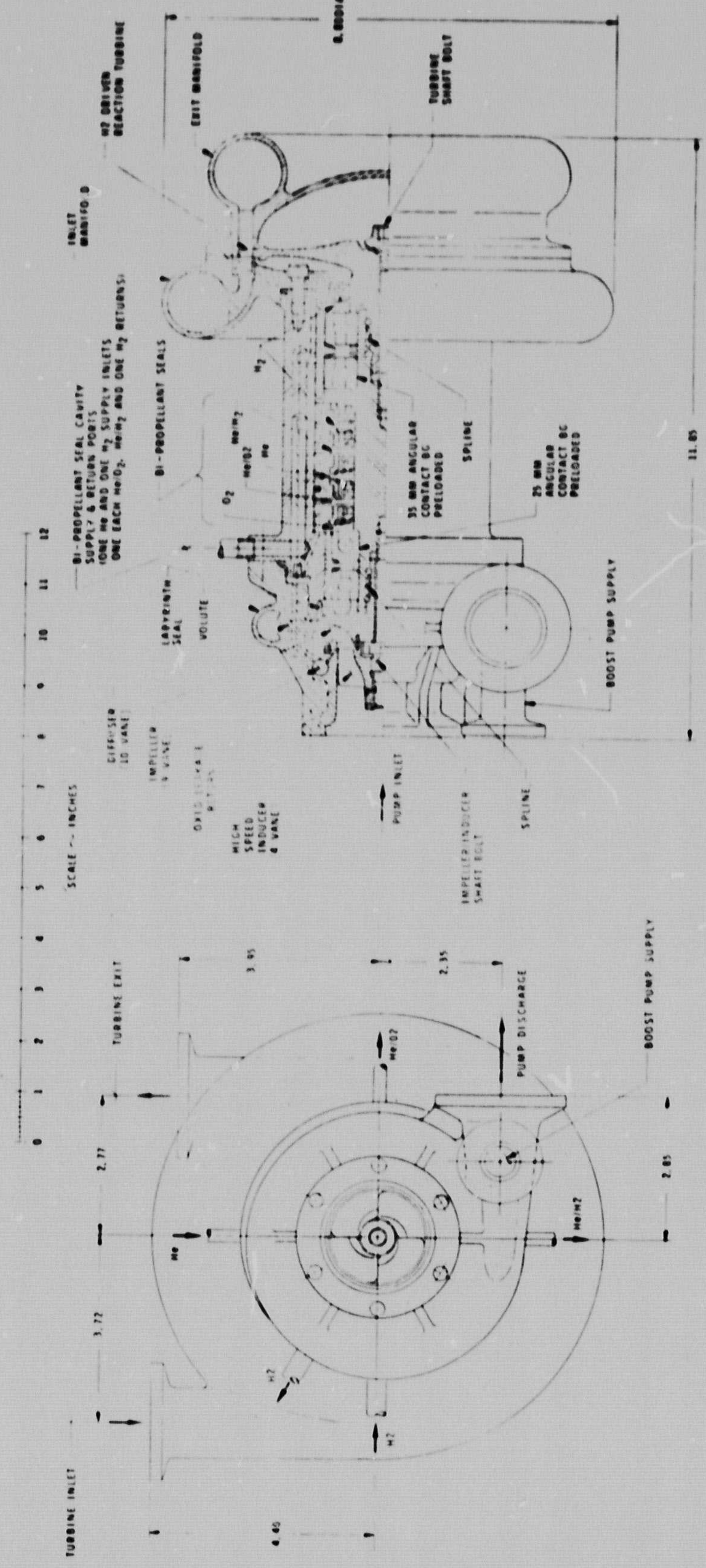

Figure 9. LO_2 TPA (ALRC Dwg. #1191999)

Helium is introduced between the second and third seal (counting from the pump end bearings) at a pressure sufficiently high so that helium leaks under both seals and mixes with oxygen on the pump side seal and hydrogen on the turbine side seal.

All static seals except the three exterior seals are of the piston ring type. The three outside seals are of the conno-seal type (AS 4061) which seal against internal pressure.

The selected turbopump shaft bearings are a pair of back to back set of deep groove angular contact at each end. These bearing sets are preloaded axially with springs so that there is no radial looseness. The turbine end set are 35mm and are locked in the housing so that one will carry axial thrust in one direction and the other in the opposite direction. The pump end set (25mm) is allowed to move axially in the housing and carries only radial load.

The turbine end bearing(s) are lubricated with liquid hydrogen. The liquid hydrogen is tapped off the second stage of the main hydrogen pump, flowed through the bearings and returned to the inlet of the main hydrogen pump second stage.

The pump end bearings take their flow from the impeller back side labyrinth leakage flow through the bearings and are returned to a manifold which feeds the flow between the pump inducer and pump impeller.

J. SHUTOFF VALVE

Series redundant main propellant valves are recommended to assure that the engine will shutdown. The design shown on Figure 10 is

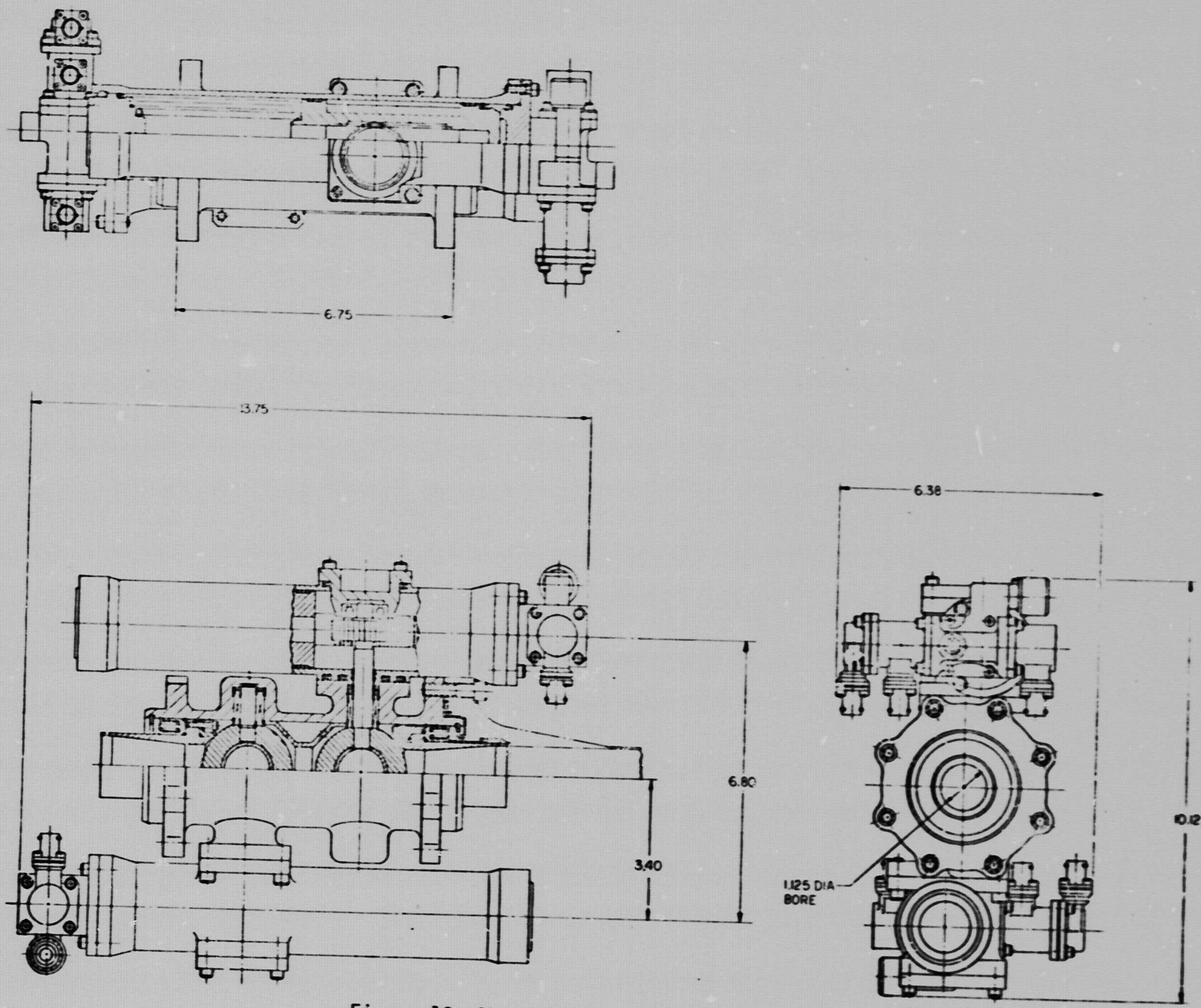

Figure 10. Shutoff Valve (ALRC Dwg. #1193175)

very similar to that used on the recently developed OMS engine. This valve design incorporates series redundant balls and actuators. The valve is pneumatically actuated to the open position and spring closed. The pilot valve, connectors and actuators can be rotated to accommodate the engine packaging and to reduce the engine envelope.

The pneumatic actuation system consists of gaseous nitrogen pressurization to provide the opening force and a spring for the closing force. This system was selected for the following reasons:

- The high pressure system minimizes the actuator size.
- The valve will fail to the closed position.
- The system is state-of-the-art and used on the OMS engine.
- No materials compatibility problems are presented.
- No additional propellant leak paths are provided.

K. MODULATING VALVE

Modulating valves are required in the line bypassing both turbines for thrust control and in the line bypassing one of the turbines for mixture ratio control. The valve design for both of these applications is shown on Figure 11. The poppet valve design uses redundant electric motor actuation which was incorporated in the design to provide a "fail-closed" capability. Other valve features are called out on the figure.

An actuator trade study was conducted considering reliability, complexity, weight, safety, control precision, and the state-of-the-art. Pneumatic, hydraulic and electric actuation systems were considered. Based upon this trade study electric motor modulating valve actuation was selected primarily to meet the control precision required for the thrust and mixture ratio control valves.

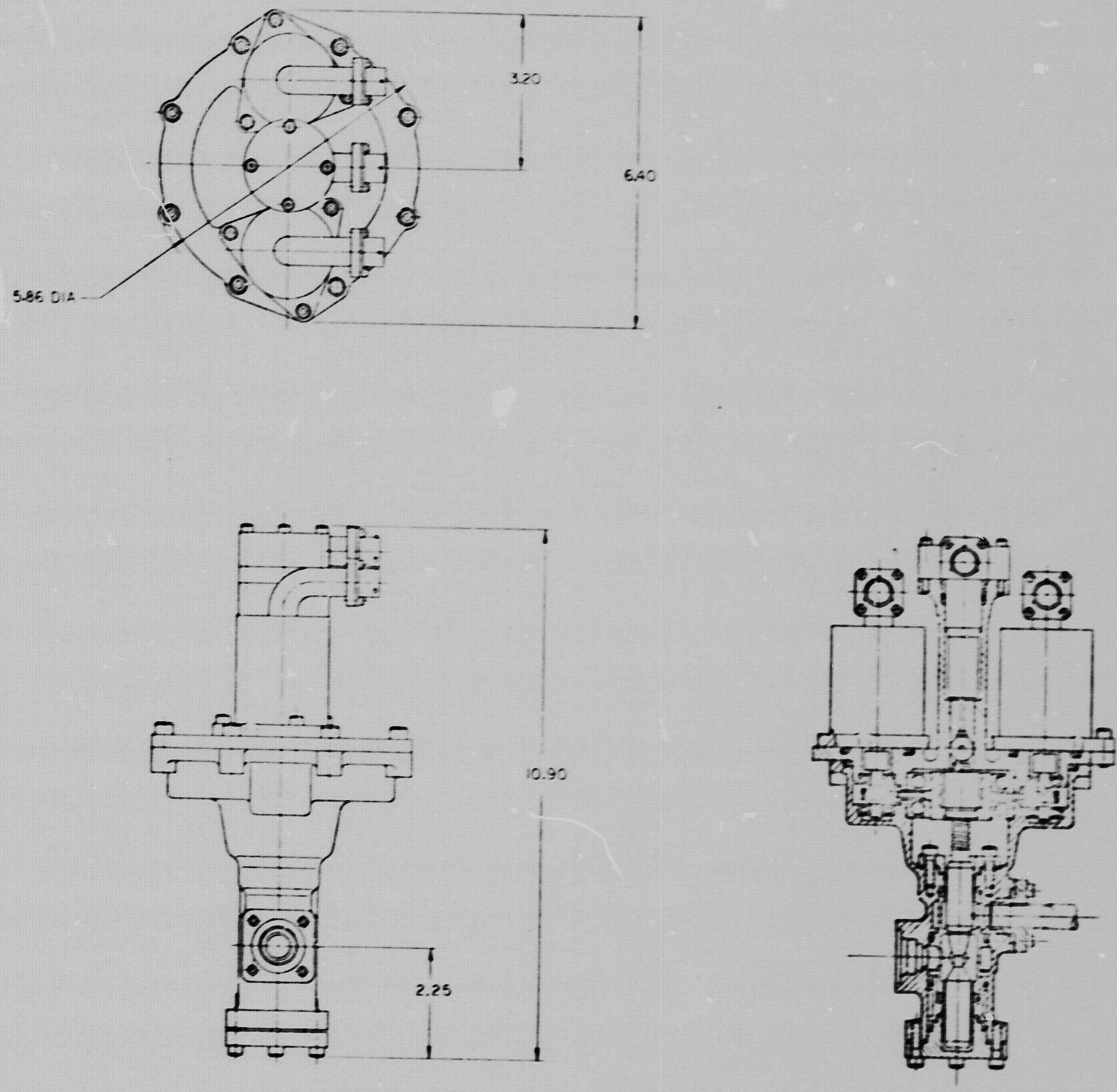

Figure 11. Modulating Valve (ALRC Dwg. #1193180)

V. ENGINE PERFORMANCE AND LIFE PREDICTION

Engine system delivered specific impulse for design and off-design operation is presented in Tables IV and V, respectively. Table V also shows the various performance efficiencies associated with design operation of the Advanced Expander Cycle Engine. These efficiencies were first calculated using simplified techniques and verified by the JANNAF rigorous performance prediction methodology (Reference 1): In addition, the experimental RL-10 and ASE performance data was analyzed and correlated by using both the rigorous and simplified JANNAF performance methodologies. The rigorous method used both the Two-Dimensional Kinetic with enthalpy addition and the BLIMP (Cebeci-Smith) boundary layer solution. The simplified model used ODK at propellant tank enthalpy with TBL-Chart/adiabatic wall conditions. While there were significant differences between the Isp_{TDK} and the boundary layer performance losses (Δ Isp_{BL}) between the two approaches, there was only 0.6 sec difference in the predicted specific impulse between the simplified and the TDK/BLIMP (Cebeci-Smith) results over a wide range of propellant type mixture ratios, chamber pressures, area ratios, and wall-temperature-to-total temperature ratios. Also, either approach predicted overall Isp's which were within approximately 0.3% of the experimental value.

The engine has been designed for 1200 thermal cycles and 10 hours accumulated run time. Therefore, all of the component designs. illustrated in Section IV of this Summary, are based on the minimum service life requirement (300 cycles or 10 hours) with a safety factor of 4 applied to lower bound data.

This service life is not predicted to be reduced when the engine is operated at mixture ratios between 6.0 and 7.0. Similarly, low thrust operation (i.e., 1500 lbF) at mixture ratios between 6.0 and 7.0 is not predicted to reduce this service life. Cooling the chamber and tube bundle nozzle was an area of concern, especially for low thrust operation of the

TABLE IV

ADVANCED EXPANDER CYCLE ENGINE BASELINE PERFORMANCE

1.	Thrust (lbf)	15000.00
2.	Chamber Pressure (psia)	1200.00
3.	Mixture Ratio	6.00
4.	Total Flowrate (lbm/sec)	31.56
5.	LOX Flowrate (lbm/sec)	27.05
6.	Fuel Flowrate (lbm/sec)	4.51
7.	I_{sp} ODE (seconds)	486.11
8.	Nozzle Efficiency	.9929
9.	Energy Release Efficiency	1.0000
10.	Kinetic Efficiency	.9957
11.	Boundary Layer Loss (lb)	164.16
12.	I_{sp} Delivered (seconds)	475.4

TABLE V

ADVANCED EXPANDER CYCLE ENGINE
PERFORMANCE AT DESIGN AND
OFF-DESIGN O/F
RATED AND LOW-THRUST OPERATION

THRUST, LB	ENGINE MIXTURE RATIO	THRUST CHAMBER PRESSURE, PSIA	ENGINE DELIVERED VACUUM SPECIFIC IMPULSE, SEC.	FLOWRATES, LB/SEC	
				Fuel	OX
15000	6.0	1200	475.4	4.51	27.05
15000	6.5	1180	474.9	4.21	27.37
15000	7.0	1162	471.0	3.98	27.87
1500	6.0	125	459.7	.466	2.80
1500	7.0	121	451.7	.415	2.91

Note: Injector elements are modified for the low-thrust condition.

engine, but thermal analysis has shown that both of these components can be designed to meet the service life requirement at both thrust levels without compromising the basic engine.

VI. ENGINE SYSTEM AND COMPONENT WEIGHTS

The Advanced Expander Cycle Engine weight breakdown is shown in Table VI. This table shows both "estimated" and "calculated" component weights. Estimated weights are based on known component weights from existing engines, and estimated component weights from "study" engines with appropriate weight scaling relationships applied to both sets of data. These weight scaling relationships are usually functions of chamber pressure, thrust and/or nozzle area ratio. Calculated component weights are based upon the component weight as derived from the component layout preliminary design drawing. Thus, calculated component weights are considered more realistic. A calculated weight for the engine controller is unavailable because the preliminary design has not been done. In this case the estimated engine controller weight is used in determining the total engine system weight.

The present component designs will be reviewed and improved in subsequent studies. Other components such as the engine controller still need preliminary design definition. Both of these activities will result in refined component and total engine system weights as the Advanced Expander Cycle Engine design matures.

TABLE VI

ADVANCED EXPANDER CYCLE ENGINE WEIGHT DATA

	COMPONENT	ESTIMATED WEIGHT, LB	CALCULATED WEIGHT, LB
1.	Gimbal	12.5	3.3
2.	Injector	16.2	30.6
3.	Chamber	48.1	47.3
4.	Copper Nozzle	20.5	27.0
5.	Tube Bundle Nozzle	46.7	38.4
6.	Rad. Nozzle	62.1	80.0
7.	Nozzle Deploy. System	47.1	72.0
8.	Valves and Actuators	59.1	72.7
9.	LOX Boost Pump	8.8	5.6
10.	LH_2 Boost Pump	3.2	8.5
11.	LOX TPA (HI SPD)	23.7	26.9
12.	LH_2 TPA (HI SPEED)	33.4	26.3
13.	Misc. Valves & Pneumatic Pack	5.2	12.6
14.	Lines	28.6	37.0
15.	Ignition System	11.0	9.2
16.	Engine Controller	35.0	35.0
17.	Miscellaneous	36.1	37.0(1)
18.	Heat Exchanger	4.8	5.0
19.	Total Engine Weight	502.1	574.4

(1) Miscellaneous includes: Electrical harness, 12.5 lbs; service lines, 6.5 lbs; TPA protective bulkhead, 0.4 lbs; attachment hardware, 15.0 lbs; and instrumentation, 2.6 lbs.

VII. ENGINE ENVELOPE

The Advanced Expander Cycle Engine envelope data are listed in Table VII.

The gimbal and injector lengths are taken from the component drawings presented in Section IV of this Summary. The chamber length of 18 inches is an optimized value based on energy release efficiency, turbine inlet temperature, pressure drop, weight and delivered engine performance. All of these parameters are strong functions of chamber length in the length constrained OTV application. Finally, nozzle length (and exit diameter) result from the required engine stowed length (60 inches).

The nozzle area ratio is the highest value possible within the constraints of engine stowed length and the selected chamber pressure. The copper nozzle and tube bundle area ratios are primarily determined by heat transfer considerations. The primary thermal consideration is to limit the maximum temperatures and temperature gradients experienced by these two components to meet the design life.

The percent bell nozzle is the result of an optimization process in which specific impulse and nozzle weight have been traded off to maximize the performance and minimize the weight within the fixed available envelope of 60" with the extendible nozzle in the stowed position.

REFERENCES

1. JANNAF Liquid Rocket Engine Performance Prediction and Evaluation Manual, CPIA Publication 246, April 1975.

TABLE VII

ADVANCED EXPANDER CYCLE ENGINE

ENVELOPE DATA

1.	Gimbal Length, in.	2.4
2.	Injector Length, in.	4.8
3.	Chamber Length, in.	18.000
4.	Total Nozzle Length, in.	84.4
5.	Radiation Cooled Nozzle Length, in.	49.6
6.	Engine Stowed Length, in.	60.000
7.	Engine Deployed Length, in.	109.6
8.	Exit Diameter, in.	58.2
9.	Throat Radius, in.	1.395
10.	Area Ratio	434.6
11.	Cu. Area Ratio	10.6
12.	Tube Bundle Area Ratio	172.
13.	Percent Bell	81.8
14.	X/RT	60.6
15.	Percent Rao	108.2